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NOTES ON THE DESIGN AND PERFORMANCE OF A THREE-STAGE ROCKET TEST VEHICLE FOR AERODYNAMIC RESEARCH AT HYPERSONIC SPEEDS

by

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This Report is one in the Series 375-397, inclusive, presenting papers, with discussions, given at the AGARD Specialists' Meeting on 'The Use of Rocket Vehicles in Flight Research' at the Kurhaus Hotel, Scheveningen, Holland, 18-21 July 1961, sponsored by the AGARD Fluid Dynamics Panel

SUMMARY

The weapons Research Establishment in Australia and the Royal Aircraft Establishment in England have jointly sponsored and developed a three-stage rocket test vehicle having as its primary function the extension of free-flight aerodynamic researches up to speeds of 10,000 ft/sec. So far, two vehicles have been flown successfully.

This report gives details of some of the design problems encountered and describes the engineering and operational aspects of the project.

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NOTES ON THE DESIGN AND PERFORMANCE OF A THREE-STAGE ROCKET TEST VEHICLE FOR AERODYNAMIC RESEARCH AT HYPERSONIC SPEEDS

J.A. Hamilton*

1. INTRODUCTION

For the past few years, the Aerodynamics Department of the Royal Aircraft Establishment has been engaged on a series of experiments using rocket test vehicles to investigate certain basic aerodynamic flow problems - notably those associated with heat transfer - at velocities up to 5000 ft/sec. Recently the question arose of extending basic free-flight work of this nature to higher speeds, the main arguments in favour being: the ever present problem of extrapolating experience gained at high supersonic speeds into the hypersonic range, primarily because of the importance of viscous effects; the difficulty, increasing with Mach number, of correctly representing in one ground facility all the relevant flow similarity parameters, and the continuing need for a test facility free of the constraints associated with ground-based equipment.

As a result of a joint study by the Royal Aircraft Establishment and the Australian Weapons Research Establishment a simple design was produced for a three-stage test vehicle based almost wholly on existing components, and having an estimated maximum velocity of 10,000 ft/sec.

This report describes some of the design and operational problems associated with the vehicle development.

2. VEHICLE DESIGN

The vehicle* has three stages: the first stabilised by fins and the second and third stabilised by flares (Figs. 1 and 2).

2.1 Propulsion

Preliminary calculations suggested that a vehicle based on three existing solid-fuel motors would achieve the required performance. These motors were designed and developed by the Ministry of Aviation's Rocket Propulsion Establishment.

The first-stage motor is a fast-burning version of the Raven, a motor on which much flight experience has been gained since it forms the propulsion unit of the high-altitude probe Skylark[1]. The fast-burning motor (burning-time 6 seconds) was chosen primarily to allow the use of a simple zero-length launcher without incurring an undue penalty in operational flexibility owing to dispersion troubles.

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*Royal Aircraft Establishment, Farnborough, Hampshire, England

*The U.K. name is 'Jaguar'.

The second and third stages are based upon well-proven motors which have been in production for some time. These two stages are ignited at about 80,000 ft altitude and, to avoid ignition troubles arising from low ambient pressures, the motors are sealed at ground-level pressure by means of diaphragms cemented into the nozzle throat. All three motors have their igniters at the head end with the appropriate leads introduced through the throat. For the first stage the igniter is of the pelleted type: for the second and third stages powder igniters are used.

2.2 Aerodynamic Design

2.2.1 Static Stability

The trajectories appropriate to the aerodynamic research programme envisaged for Jaguar all follow a similar pattern (Figs. 3 and 4): first-stage boost to a velocity of 3000 ft/sec at an altitude of 7,000 ft followed by a period of coasting to an apogee of about 90,000 ft altitude; the second and third stages are then fired in quick succession when the flight path is between 10° and 15° below the horizontal (Fig. 3). Consideration of heating and aerodynamic effectiveness problems within these flight régimes led to the adoption of fin stabilisation for stage one, and flare stabilisation for stages two and three.

The area of the first-stage fin assembly is such as to ensure adequate stability for the complete vehicle, even allowing for uncertainties in the methods of estimation (Fig. 5). For the flare-stabilised stages however, the corresponding methods of estimating static stability appeared to be less well established, particularly for the high-fineness-ratio bodies appropriate to Jaguar. Two methods were used for stability analysis; that of Phythian and Dommett[2] and that of Syvertson and Dennis[3]. The differences between the centre-of-pressure and lift-curve-slope values given by these two are shown in Figure 6 for stages two and three together, and in Figure 7 for stage three alone: for the higher Mach numbers the Syvertson and Dennis method gives a lower lift-curve slope and a greater stability than that due to Phythian and Dommett. The major source of this disagreement appears to lie in the amount of lift induced on the parallel components by nose cones and flares. In order to try and establish which estimate was more appropriate to the present vehicle, a number of wind-tunnel measurements were made on the flare-stabilised stages.

Stages two and three in combination were tested at a Mach number of 4.3 in the 11 in × 6 in wind tunnel at the Royal Aircraft Establishment: stage three alone was tested in the 15 in × 15 in wind tunnel at the Weapons Research Establishment over a Mach number range of 1.8 to 2.8 and at $M = 4.3$ in the R.A.E. 11 in × 6 in wind tunnel. Insofar as one can generalize about the results they suggest that for centre-of-pressure position estimates the Phythian and Dommett analysis gives the better agreement with experiment (Figs. 6 and 7) and, since the Phythian and Dommett results are the more pessimistic, their values were used for design purposes.

The estimation of stability is further complicated by the non-linear pitching-moment/incidence relationship which is characteristic of some flare-stabilised bodies. On the third stage of Jaguar, for example, this took the form of a marked reduction in stability over the incidence range $\pm 2^{\circ}$ (Fig. 8): such a variation is almost certainly associated with shockwave/boundary-layer interactions at the cylinder/flare junction and may be highly sensitive to Reynolds number. However since the tunnel

R.N. (4×10^6) differs from full scale (12×10^6) by a factor of three only, one cannot rely on the effect disappearing under flight conditions. The comparisons of c.p. position made earlier are based on the region outside the $\pm 2^\circ$ incidence band.

2.2.2 *Dynamic Stability*

During flights of the high-altitude probe Skylark[1] there have been occasional instances of large-amplitude yawing motions occurring at altitudes around 100,000 ft. A qualitative explanation for these motions has been found in terms of roll/yaw interactions and, although there are manifest differences in the flight plans of the two vehicles, the experience with Skylark prompted an investigation into the effect of roll on the dynamic stability of Jaguar. The main points of this investigation are summarised in a paper by Shannon[4].

So far as the classical case of resonance between yaw and roll frequencies is concerned, the situation for Jaguar is illustrated in Fig. 9 (taken from Reference 4), which compares the estimated rates of roll for various fin cant angles with the estimated natural frequency in yaw. In practice, fin misalignments are likely to be of the order of 0.1° and it is clear from Figure 9 that, if there is a danger from roll/yaw resonance instability, it is likely to arise towards the end of first-stage coast (flight time 60-70 seconds).

By stringent control of fin misalignments one ought to be able to keep the roll frequency below the yaw natural frequency at all times and this is the procedure which was adopted for the first two vehicles flown* (see also Section 5). An alternative solution is to introduce a deliberate amount of fin cant such that the roll frequency is always well above the yaw frequency.

2.2.3 *Drag*

Estimated values of the zero-lift drag coefficients for the various components of the vehicle are given in Figure 10.

2.3 Structural Design

The three-panel fin assembly for the first stage (Fig. 11) is similar to that used on Skylark[1], the principal differences being the use of a steel outer skin and leading edge in place of light alloy. Although the light-alloy skin is probably adequate in terms of strength, the change to steel was thought wise in order to minimise thermal distortion and its possible repercussions on roll behaviour.

The junction between stages one and two consists of a simple spigoted joint (Fig. 12): the spigot itself is hollow and is suitably vented to atmosphere in order to minimise blockage effects in the motor nozzle immediately after second-stage ignition. The design criterion adopted for the spigot and nozzle was that they should be capable of withstanding the bending loads required to fail the second-stage motor tube at its aft end plate. A conventional skin and stringer construction of light alloy was adopted for the second-stage flare.

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*In this context one should note that the estimated rates of roll quoted in Figure 10 are approximate only, and intended to indicate no more than a general trend.

For the coupling between stages two and three a diaphragm joint of the kind originally developed by N.A.S.A. was employed: this consists of a light-alloy disc 11 in diameter, threaded on its outer rim and having a series of radial slots and an annular groove milled into the face (Figs. 13 and 14). The diaphragm is screwed halfway into a mating thread in a steel transition piece fitted to the head end of the stage-two motor and halfway into another mating thread in an extended nozzle fitted to the third stage (Fig. 13). This extended nozzle also acts as the stabilising flare for the third stage.

The diaphragm thickness was somewhat arbitrarily chosen as 0.20 in, this being regarded as a suitable compromise between achieving adequate strength and stiffness without incurring ill effects on the third stage at ignition, owing to excessive inertia of the diaphragm. The depth of the annular groove was chosen to give separation between the stages when a static pressure of about 50 lb/in^2 was applied to the face of the diaphragm. Again this figure was a compromise between avoiding premature separation, owing to high transient accelerations during stage-one and stage-two burning, and avoiding an undue influence of the diaphragm on the pressure surge at final-stage-motor ignition. The thickness of metal at the bottom of the groove (0.10 in) was determined by experiment.

Ground tests on this coupling proved that, while the system functioned satisfactorily, the pressure just ahead of the diaphragm rose to about 220 lb/in^2 after 2 milliseconds and then fell away linearly to nearly zero at 9 milliseconds. This pressure variation would correspond to high transient longitudinal accelerations of about 200g and attempts were made to improve the rearward movement of the diaphragm and the venting in the transition section. Even with these improvements the transient accelerations are estimated to be very high, but they appear to have had no ill effect on the vehicle or its instrument installation.

The structural design of the third stage was dominated by the problem of achieving adequate stiffness rather than the provision of adequate strength. Although the short burning time of the first-stage motor had advantages in terms of reducing dispersion, it also implied a combination of high velocity and low altitude at the end of first-stage burning.

As describes in the Appendix, a simple static analysis was made of the aeroelastic characteristics. The measure of elastic deformation produced by this analysis has been called the 'nose-incidence magnification ratio'. In simple terms it is the ratio between the trimmed incidence of the final stage and the initial static misalignments between the stages. An absolute upper limit of 10 was given to this ratio; with initial misalignments of about 0.5° this implies a final-stage incidence of 5°. Typical altitude speed boundaries for ratios of 10 and infinity are illustrated in Figure 15.

For the vehicle as originally designed, the estimated divergence speed was well below maximum first-stage velocity. The principal source of trouble was the flexibility of the final stage and this was therefore stiffened by a steel sleeve (which now combines the duties of stiffening sleeve, heat shield and antenna). Even with this improvement, and the addition of ballast to reduce the maximum first-stage velocity, the vehicle was uncomfortably near the estimated divergence curve (point marked Jaguar 1 on Figure 15).

Since the solution of this problem was fundamental to the success of the whole project, a dummy three-stage vehicle was assembled having a live first-stage and inert second-and third-stages. The stiffness characteristics of this trial vehicle were slightly better than those of the prototype three-stage (broken curves of Figure 15) but the flight plan implied that the vehicle was well into the estimated divergence region at the end of first-stage boost (point marked Jaguar G-1 on Figure 15). In spite of this the flight was wholly without sign of incipient divergence, as was the flight of the first three-stage vehicle, Jaguar 1.

The reason for this discrepancy between estimate and experiment is not yet fully established but unpublished work by Taylor of the Royal Aircraft Establishment indicates that, qualitatively at least, better agreement may be achieved if dynamic response effects are included.

2.4 Thermal Protection

Two criteria had to be satisfied in deciding the amount and character of the protection against aerodynamic heating: first that the temperature of load-carrying structures should be low enough to justify the strength and stiffness values assumed in the design and, second, that no damage should be done to the rocket-motor charges because of high temperature. The critical factor controlling the second criterion is possible failure of the bonding between charge and motor case: the quality of the bond deteriorates with increasing temperature, becoming uncertain above 60 to 70°C. These temperatures are by no means well established - little experimental evidence exists on the effect - and the Rocket Propulsion Establishment regard them as somewhat pessimistic.

The flight plan of the vehicle is such that no thermal protection is necessary for stage one. For stage two, protection was provided by Durestos* coverings applied to the flare and motor exterior. Estimates confirmed that, with a thickness of 0.150 in of Durestos, the motor wall temperature could be kept below 90°C, even with extreme ground-level ambient temperatures (Fig. 16). For the light-alloy flare a maximum temperature of 110°C was specified and, to maintain this limit, 0.120 in of Durestos proved to be adequate (Fig. 17).

Since an external sleeve was needed on the third stage to ensure adequate stiffness it was an obvious move to make this serve also as a heat shield: additional protection was provided by a cork filling between the sleeve and the motor wall (Fig. 18). The steel nozzle-cum-stabilising-flare of the third stage needed no external protection.

2.5 Antenna Design

Details of the instrument installation used in Jaguar will vary from round to round, but all rounds have common requirements in terms of transmission and reception for antennae. These are as follows:-

For telemetry	Transmit at 465 Mc/s
For Doppler	Receive at 104 Mc/s
	Transmit at 208 Mc/s
For command ignition	Receive at 166 Mc/s

*Durestos is a proprietary product consisting of an asbestos felt impregnated with phenolic resin.

The 465 Mc/s telemetry signal is transmitted by exciting the nose of the instrumentation compartment (Figs. 1 and 19). In order to reduce asymmetries in the polar pattern as the round rolls, the nose is fed at three points 120° apart: this arrangement has the additional benefit that it allows impedance matching to be obtained by using the three leads effectively as a quarter-wave transformer. Reception and transmission of the Doppler-transponder signals is achieved by exciting the stiffening sleeve on the final stage (Figs. 1 and 18): the 104 Mc/s and 208 Mc/s signals are separated by means of aerial-stub arrangements.

Simple spike aerials mounted on the spigot between stages one and two are used to receive the command-ignition signal for stage two.

2.6 Launcher Design

The launcher is of a zero-length type with a single beam offset 5 ft from the centre line of the turret support (Fig. 20). This design was chosen for three main reasons:-

(a) It avoids, as far as possible, rocket blast loads on the launcher structure.

(b) It enables most of the launcher drill to be done with the missile horizontal and only 5 ft above the launcher apron.

(c) It is a simple, straightforward design which can be readily stressed.

The main structure consists of a trussed, steel beam attached to one end of a large steel tube cantilever which is supported by two bearings, thus enabling it to rotate about a horizontal axis and provide elevation training over the range -2° to 88°. The bearings, in turn, are mounted in a heavy steel gear box which is rigidly attached to another large steel tube supported vertically by two more bearings at the top and bottom of the turret base structure, thus providing training in azimuth over $\pm 45^{\circ}$. The turret base is securely bolted to a massive, reinforced concrete foundation, which is protected by heavy sheet steel where the rocket jet impinges on its surface.

Elevation training is effected by a motorised screw jack actuating a lever attached to the horizontal tube supporting the launcher beam. Azimuth training is achieved by a hand-operated drive between the turret base and the steel box structure on top of the vertical tube. The QE of launch is determined by reading a calibrated mechanical revolution counter connected to a suitable point in the mechanical transmission.

The test vehicle is suspended beneath the launcher beam by two, loose-fitting hooks and eyes, one set at each end of the stage 1 motor. The forward hook which is attached to the launcher beam, is 'split-in-half' vertically and fitted with torsion springs so that it will quickly retract at launch and thus avoid interference as the vehicle moves past it. To reduce still further the possibility of fouling, the launcher beam has been made triangular in section, with the apex of the triangle downwards.

Roll support for the vehicle on the launcher is achieved by two triangulated outriggers mounted from the rear of the launcher beam. At the end of each of these outriggers is a rubber-tyred wheel which bears on the stage 1 stabilising fins near their outer edges. Each outrigger is fitted with a screwed adjustable link to facilitate test vehicle loading and to cope with both triple and cruciform fin configurations.

3. OPERATION

3.1 Flight Plan

For general aerodynamic experiments the aim is to have the final stage achieve maximum velocity at altitudes between 70,000 and 100,000 ft on a trajectory which is inclined earthwards at 10-15^{o} to the horizontal (Fig. 3). First-stage boost lasts about 6 seconds and is followed by a coasting phase which continues until the vehicle has passed its apogee and is on the required flight path. At this point the second-stage is fired by command from the ground: the third-stage is ignited automatically by an inertia switch as soon as the acceleration has fallen to a prescribed value.

Real-time plots of height versus time and lateral displacement versus time are provided throughout the flight in order to check whether it is safe to fire stage two and to give the operator an indication of the appropriate instant to fire.

Command ignition of stage two was chosen because for this vehicle it provides a more effective safety device than command destruction, and because it allows in-flight adjustments to the flight path to accommodate deviations from the predicted path during first-stage burning and the subsequent coasting period.

The detailed ignition sequence is as follows:-

(a) First-stage ignited by tail leads from range control. During first-stage acceleration the second-stage firing circuit is armed by an inertia switch.

(b) When the vehicle has coasted beyond apogee, the second-stage is fired by a command signal transmitted from range control to a receiver carried within the spigot between stages one and two.

The third-stage firing circuit is armed by an inertia switch during second-stage burning.

(c) When the acceleration of stages two and three combined has fallen to a prescribed value, a double-acting inertia switch fires the final stage.

3.2 Tracking

During first-stage boost and the subsequent coasting period, trajectory data are derived mainly from optical sources (kinetheodolites) and radar. Since the drag weight ratios of stage one, and stages two and three combined, are relatively close, a simple system of connecting pins is provided at the stage one and two interconnections to ensure that the vehicle stays intact until second-stage ignition, thus easing the problem of optical tracking.

After second-stage ignition the primary source of trajectory data is the range multi-station-Doppler system, using a network of ground stations and a transponder unit carried in the final-stage instrument bay.

3.3 Dispersion Estimates

The vehicle is assumed to roll at such a rate that structural and thrust misalignments produce negligible dispersion. The important parameters governing dispersion are then cross winds, steady and unsteady, separation disturbances, and height and attitude errors of the vehicle at stage-two ignition.

3.3.1 Wind and Separation Effects

First boost phase. The change in heading of a thrusting vehicle entering a cross wind which lasts for a prescribed time can be computed by standard methods giving the angular dispersion due to the wind. For launcher corrections the nominal wind pattern is divided into height bands, in each of which the wind is considered constant in speed and direction. By using the wind components normal to the still-air trajectory and parallel to the earth, the change in heading angle in the azimuth plane as the vehicle passes through each height band can be computed. The boost phase covers only a small portion of the overall flight path and it is sufficient to consider all heading changes as occurring at the launcher to obtain a launcher correction on bearing.

By considering wind components normal to the vehicle still-air trajectory and in the plane normal to the earth, corrections to the launcher elevation can be found.

First coast phase. Lateral drift due to wind in coast phases is assumed negligible. However, after stage-one burn-out, cross winds will alter the flight-path angle and at stage-two light-up the vehicle could be heading off course. Assuming the vehicle always points into the relative wind, the heading angle in azimuth at stage-two ignition can be found. The error in impact due to this cause can be considered as being equivalent to a launcher bearing error, which can readily be computed*. Errors in launcher elevation have not been considered for this case.

Second and third boost phases. Using an analysis similar to that for the first boost phase, an additional heading error can be computed and the launcher bearing can be adjusted to correct for this. The error in elevation is negligible since the vehicle is flying approximately horizontally.

Errors in final coast phase. Launcher corrections have not been considered during this part of the flight.

By assuming a constant cross wind of 10 ft/sec the errors arising from all boost phases have been computed using the asymptotic response in heading to a cross wind, and the error in impact point assessed. To this has been added the estimated error due to a poor separation at ignition of stages two and three. It has been assumed that separation could lead to the second stage heading 3^{o} off course in any direction.

*The fact that stage-two ignition occurs about 20 miles down range must be taken into account in determining the effective launcher error.

While meteorological data are expected to have errors of ±10 ft/sec in wind, this may not be representative of the gustiness prevailing at ground level, which could have a large effect on lateral dispersion since the aerodynamic restoring moments on the vehicle are small just after launch. The sum of these two sources of lateral dispersion has therefore been multiplied by a factor of 3 to allow for gusts near ground level.

Table I summarises the contributions of these various components to the lateral dispersion.

3.3.2 Longitudinal Dispersion

Longitudinal dispersion is linked with the technique for firing stage two. The vehicle is tracked from launch and the flight trajectory is displayed on plotting tables. By matching the trajectory with a set of pre-computed curves, the instant of ignition of stage two can be determined to cause the third stage to impact close to a pre-selected range. By allowing some tolerance in this range the height of stage three can be controlled to some extent, thus giving a better controlled experiment. At the instant of curve matching, stage two is fired by a command signal from the ground. From the errors in the plotting table information, the estimated error in matching the trajectory with the pre-computed curves, and the assumption that the second stage could fly 3° off course, the longitudinal dispersion of each stage can be computed.

4. PERFORMANCE

The estimated variation of maximum velocity and altitude is illustrated in Figure 21 for various first-stage configurations. These estimates are all based on trajectories of the pattern shown in Figure 3. The vehicle should be capable of a maximum velocity of about 10,000 ft/sec at an altitude around 90,000 ft. An increase in the burning time of the first-stage - from 6 seconds to 15 seconds - appears to have little influence on the maximum-velocity/altitude relationship: such an increase would ease the divergence problem, but would also reduce the operational flexibility of the vehicle owing to increased dispersion.

5. INITIAL FLIGHT TRIALS

Two test vehicles have been flown so far. The first of these had a heavily-ballasted first-stage (600 lb) in order to ensure that no divergence occurred at the end of first-stage burning; on the second vehicle the ballast was halved. On the whole, both trials were successful. The first vehicle achieved a maximum velocity of 9200 ft/sec at an altitude of 76,000 ft: this velocity is lower than was predicted by about 200 ft/sec. Most of this discrepancy could be accounted for by the fact that the third-stage ignited somewhat late for optimum performance. The second vehicle achieved a maximum velocity of about 9600 ft/sec at 97,000 ft altitude. Good telemetry and Doppler signals were received from both vehicles for about two minutes.

The major untoward incidents were the occurrence of a rolling motion during and after second/third-stage boost (the roll rate in the second vehicle was about 0.3 rev/sec) and the existence of an appreciable yawing motion on the second vehicle during the period before second-stage ignition. Neither of these incidents appears to have affected the ultimate performance of the vehicle, but the presence of the yawing motion is disturbing in terms of possible large dispersions in final-stage trajectory. Whether or not the motion is associated with a roll/yaw phenomenon is not yet clear: the rates of roll during first-stage coast are very small. Various palliatives, and monitoring devices are being investigated.

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TABLE I

Lateral Dispersion

Stage-three impact range is assumed to be at 90 miles and a constant cross wind of 10 ft/sec is assumed.

Event	*Dispersion angle (rad/unit wind)*	*Range to impact (miles)*	*Dispersion of stage three (miles)*
First boost	0.0033	90	2.9
Second boost	0.0011	70	0.8
Third boost	0.0013	70	0.9
Coast	Assumed negligible		
Poor stage separation	3° heading angle change	70	3.6
			Total 7.2 miles

Allowing for ground-level gusts, total lateral dispersion = 22 miles.

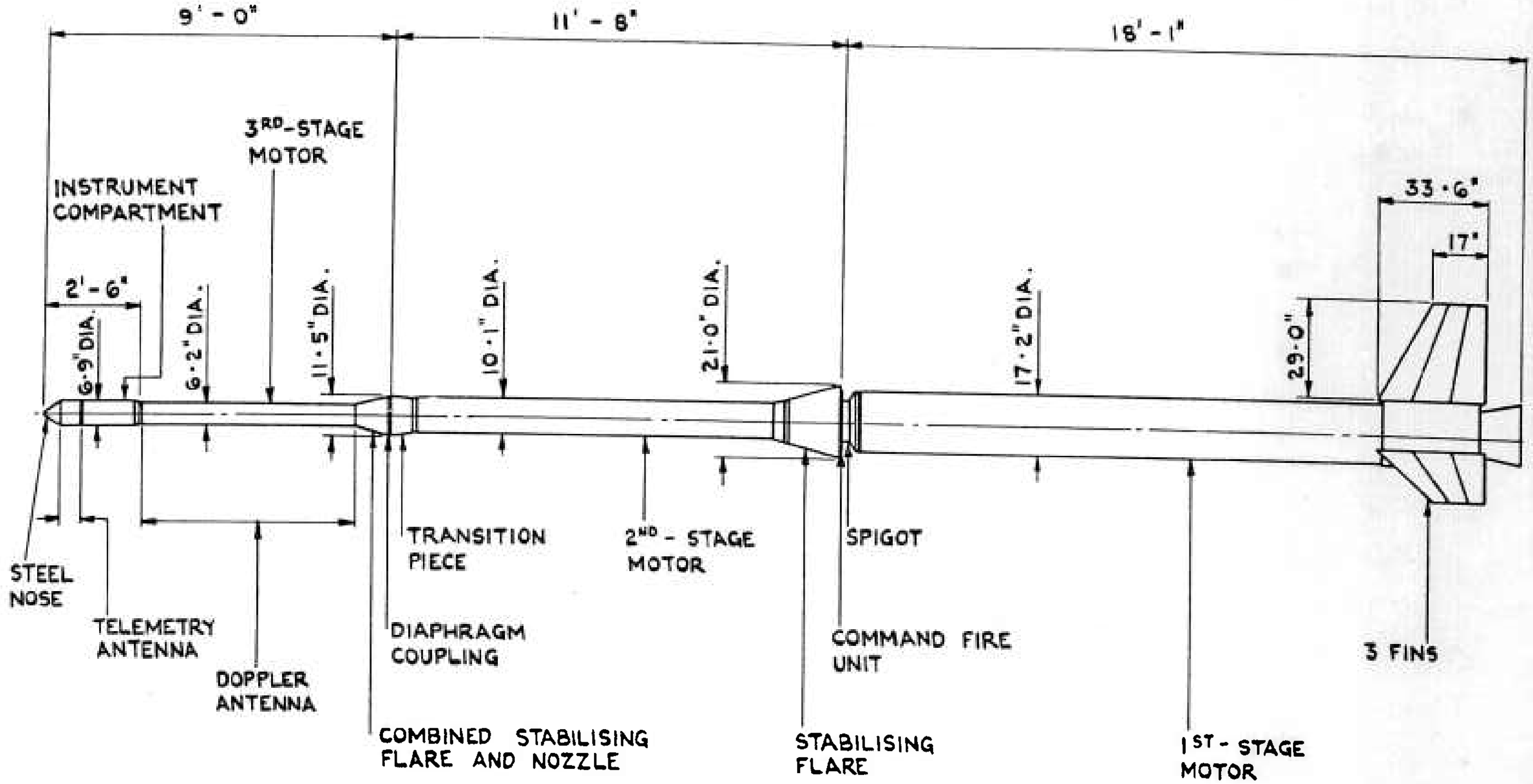


Fig.1 The hypersonic test vehicle jaguar

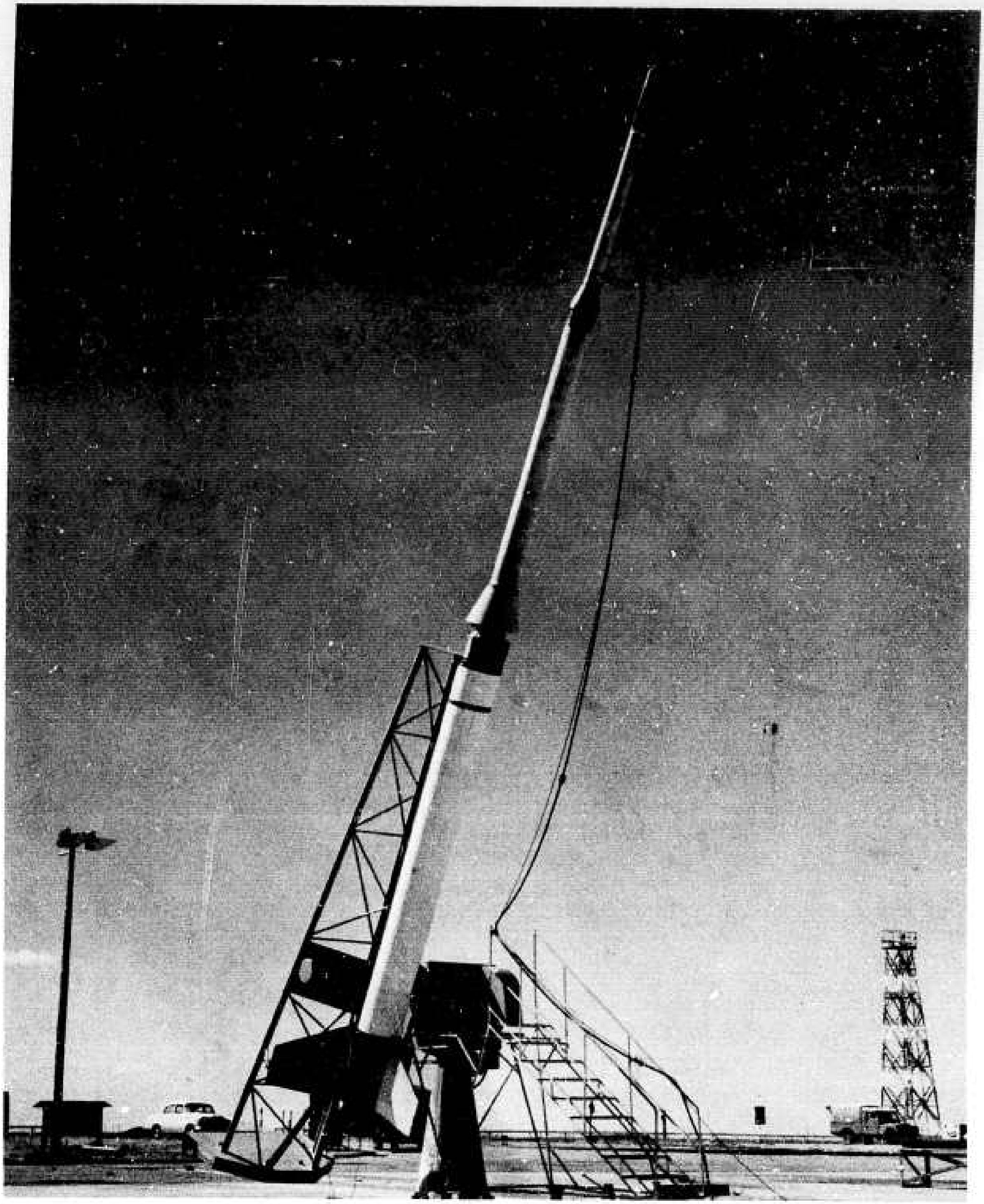


Fig.2 Test vehicle and launcher

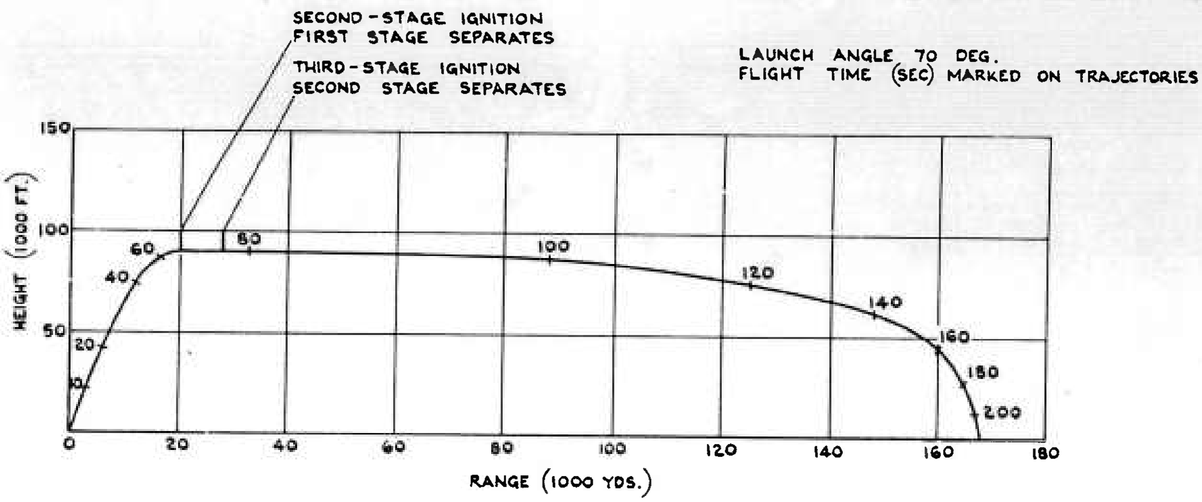


Fig.3 Typical trajectory (estimated)

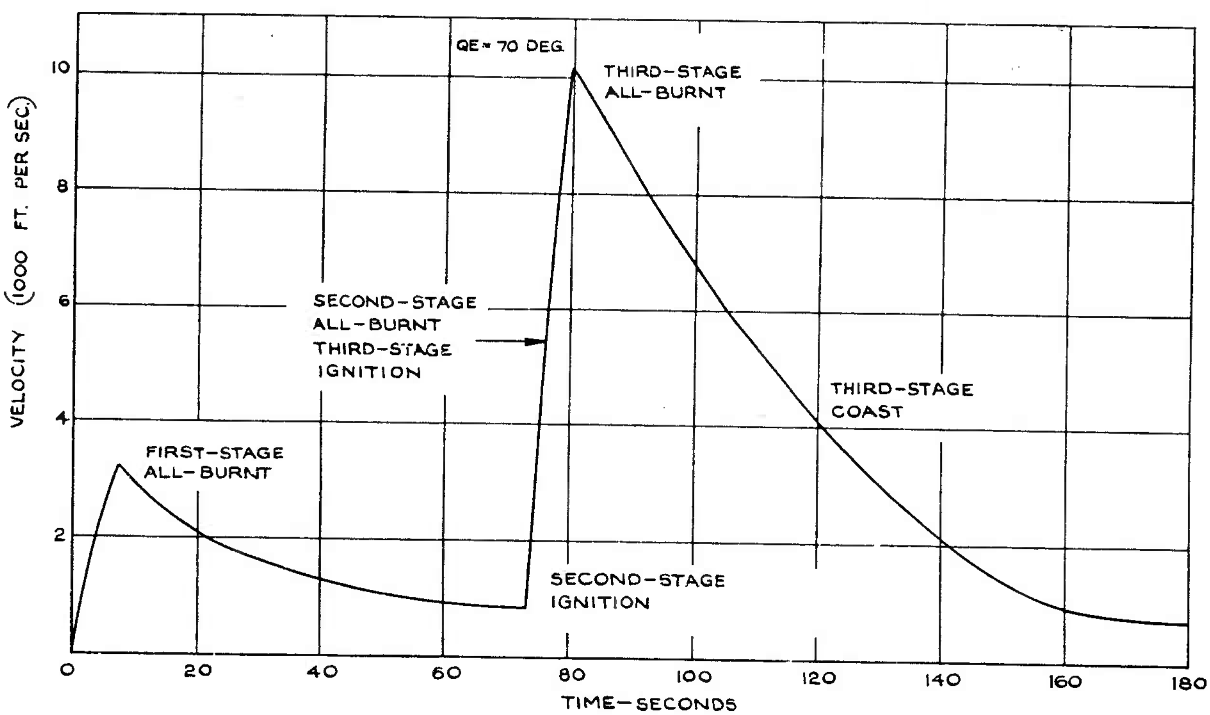
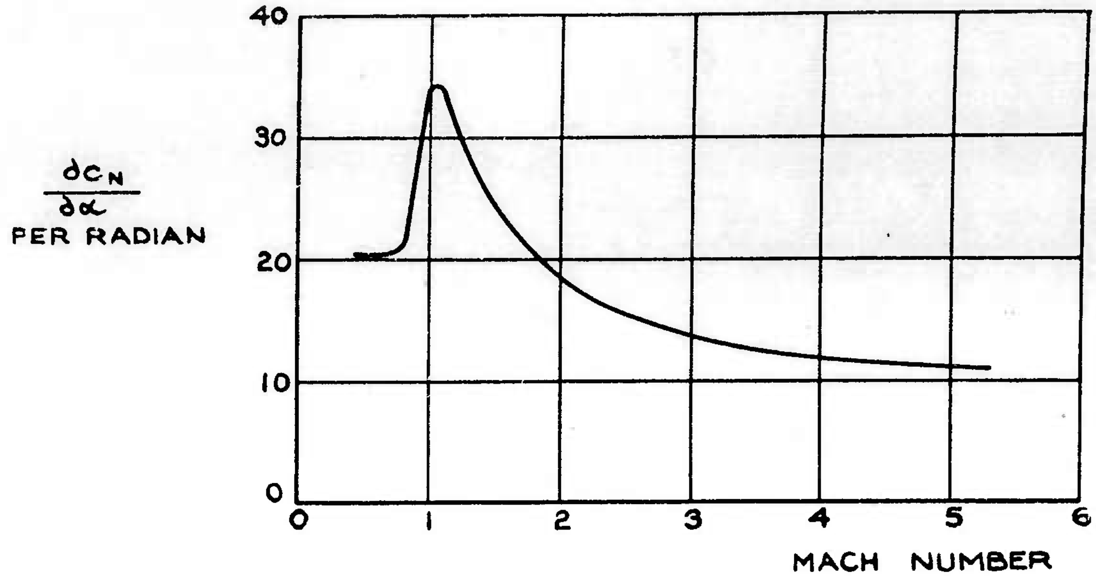


Fig.4 Estimated velocity/time curves



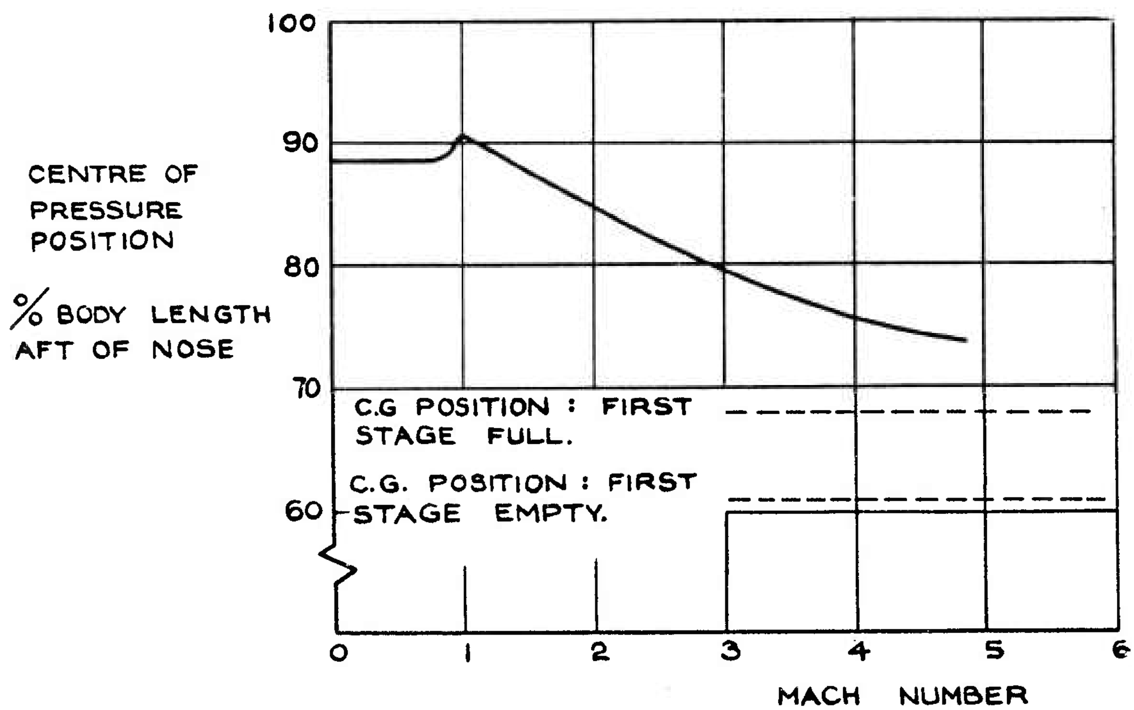


Fig.5 Estimated stability at zero incidence of stages 1, 2 and 3 in combination

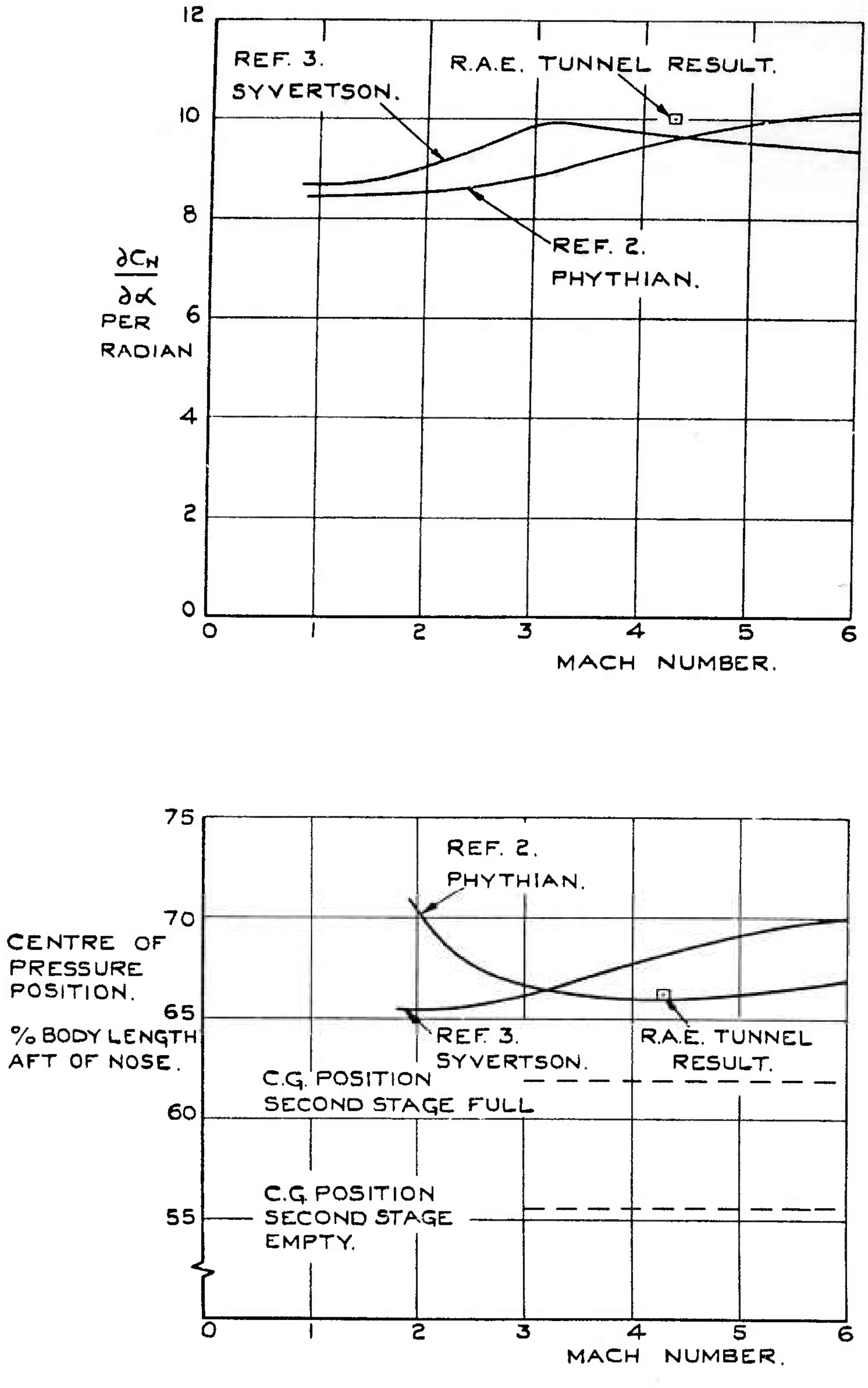


Fig.6 Static stability at zero incidence of stages 2 and 3 in combination

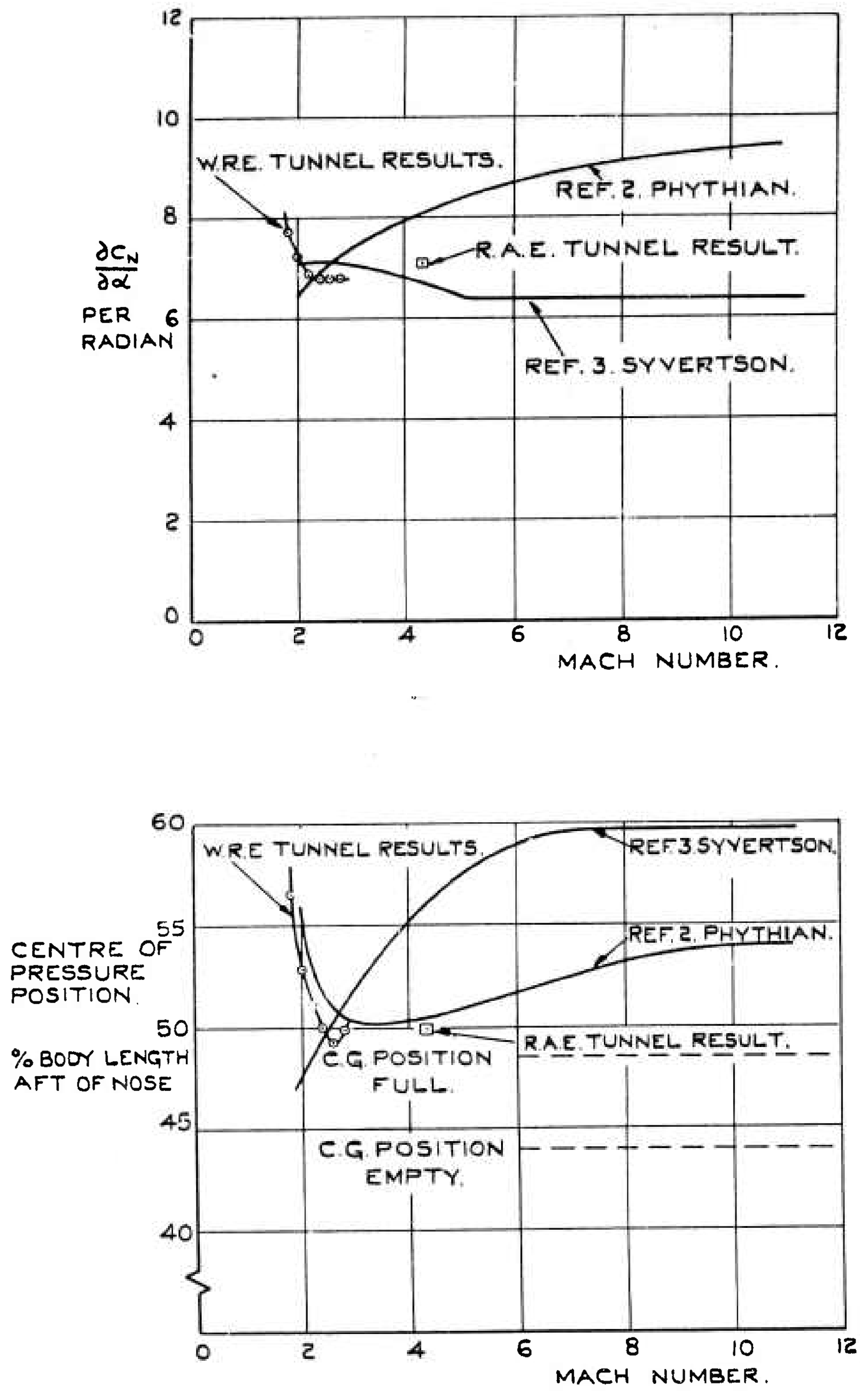


Fig.7 Static stability at zero incidence of stage 3

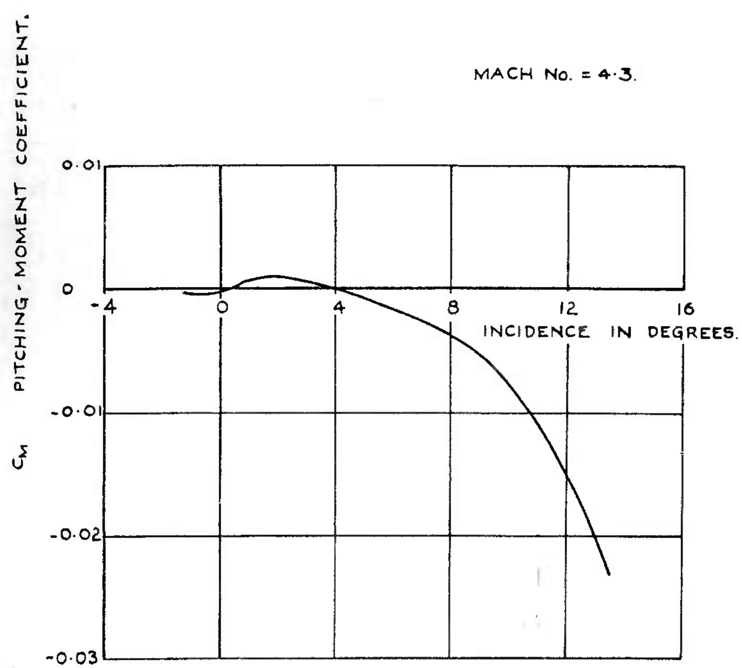


Fig.8 Measured variation of pitching moment with incidence for stage 3

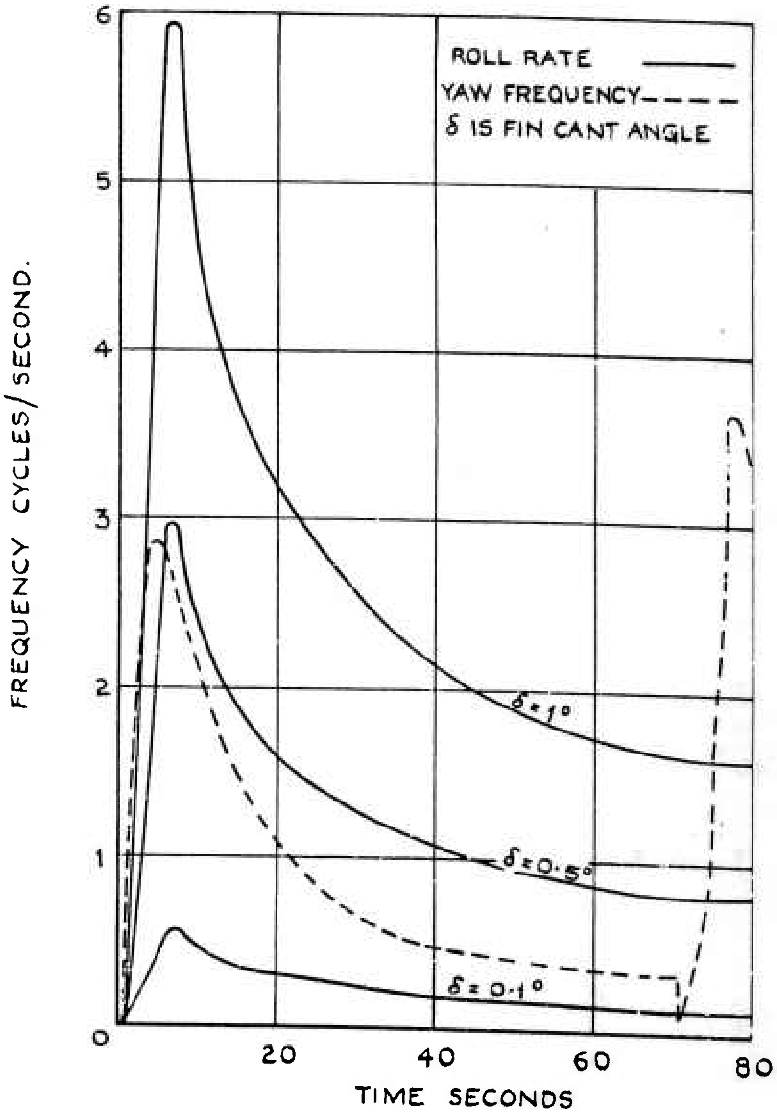


Fig.9 Estimated variation of yaw and roll frequencies with time

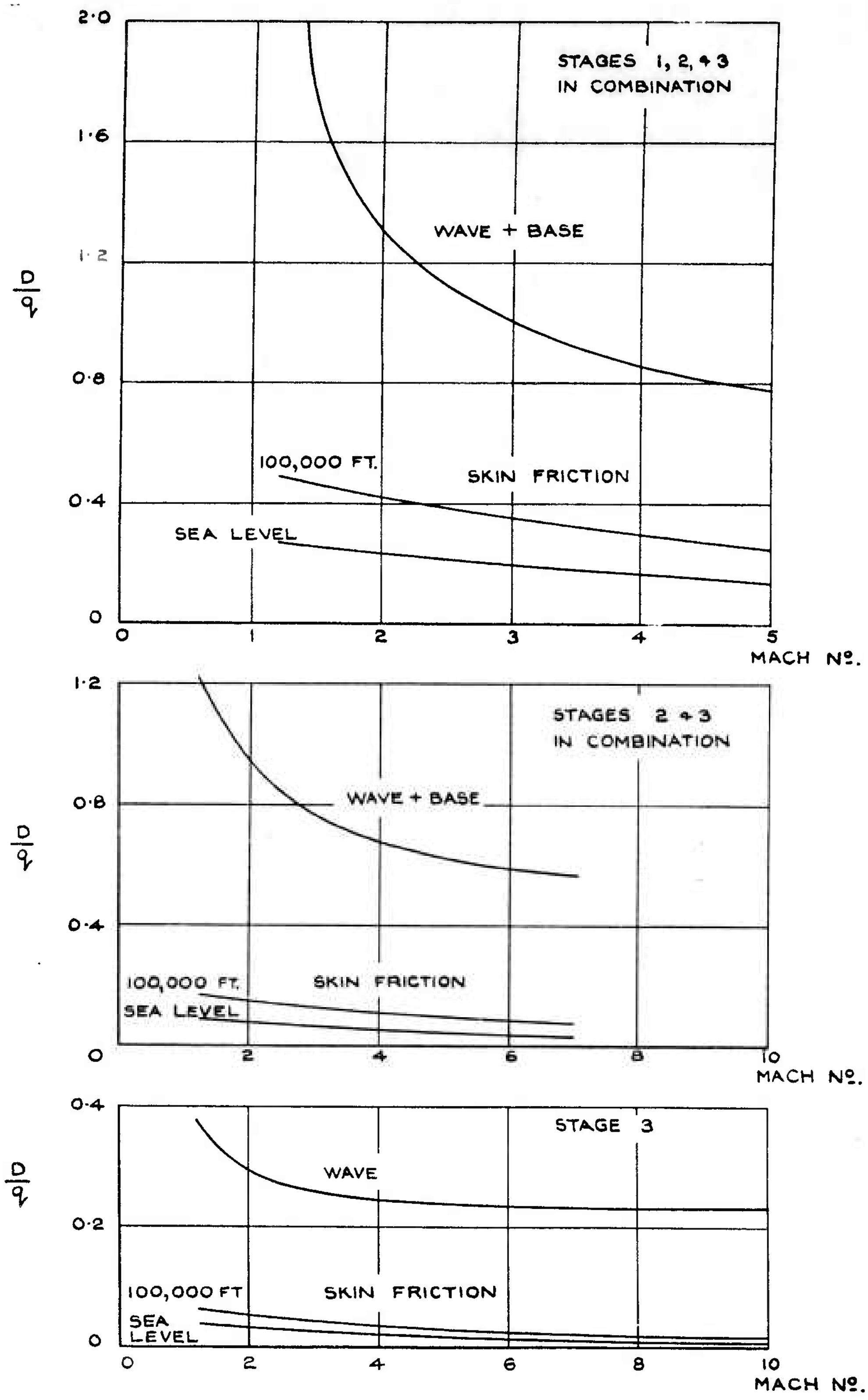


Fig. 10 Estimated zero-lift drag

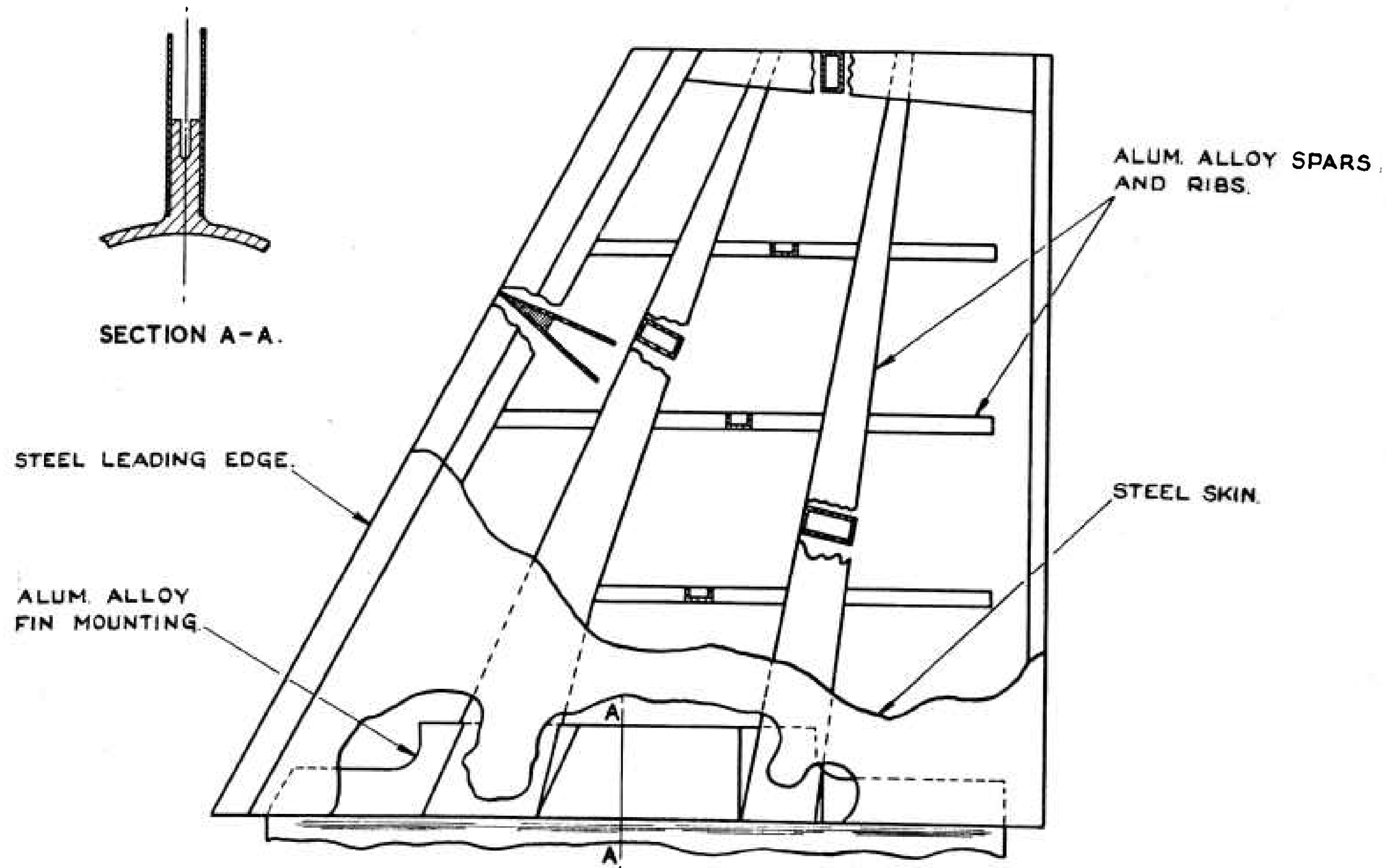


Fig.11 Details of first-stage fin

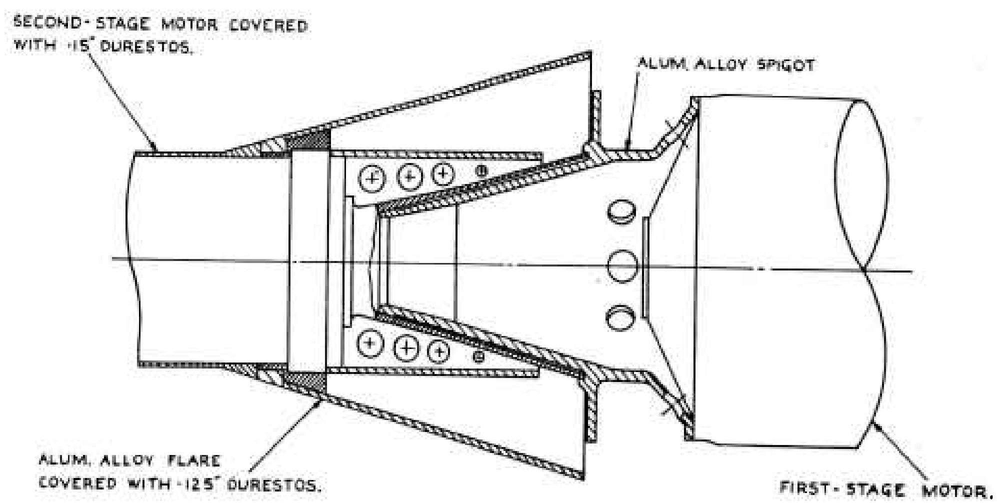


Fig.12 First/second-stage coupling

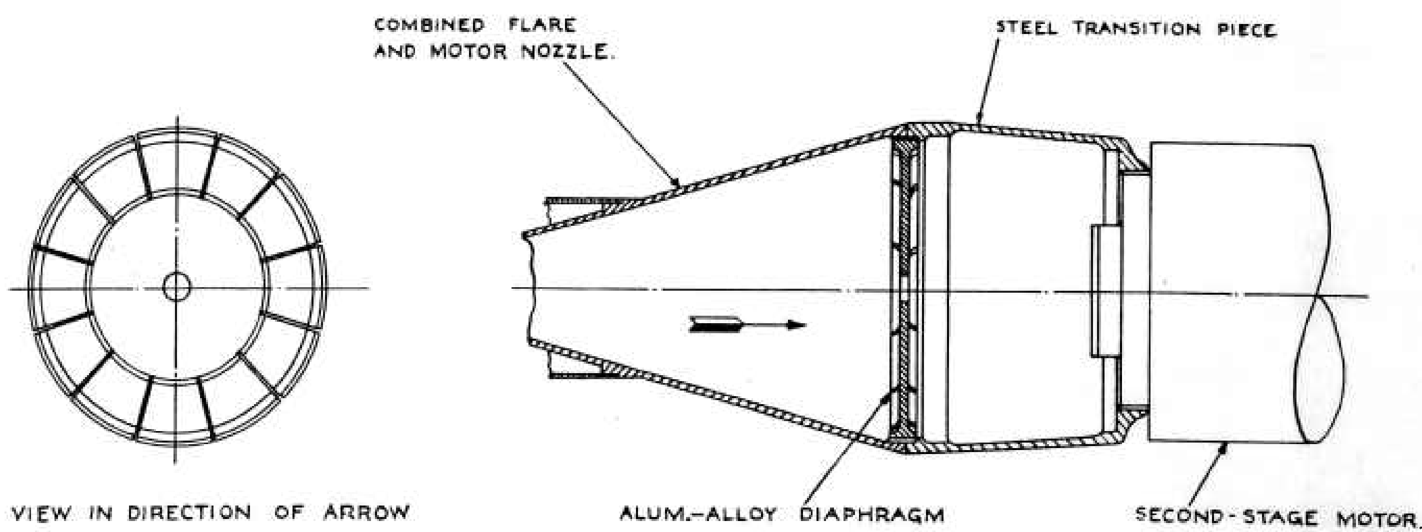


Fig.13 Second/third-stage coupling

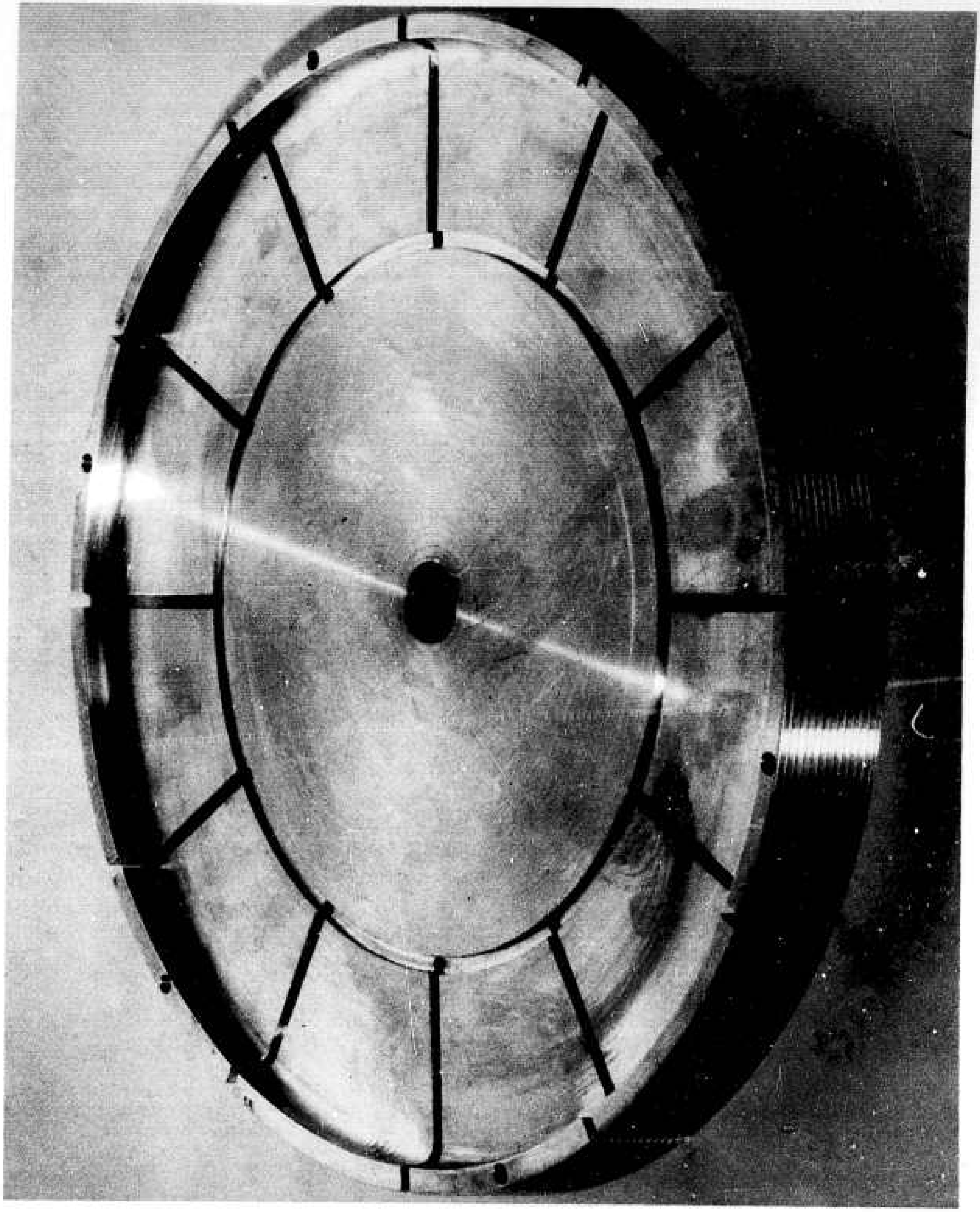


Fig.14 Aluminium-alloy diaphragm

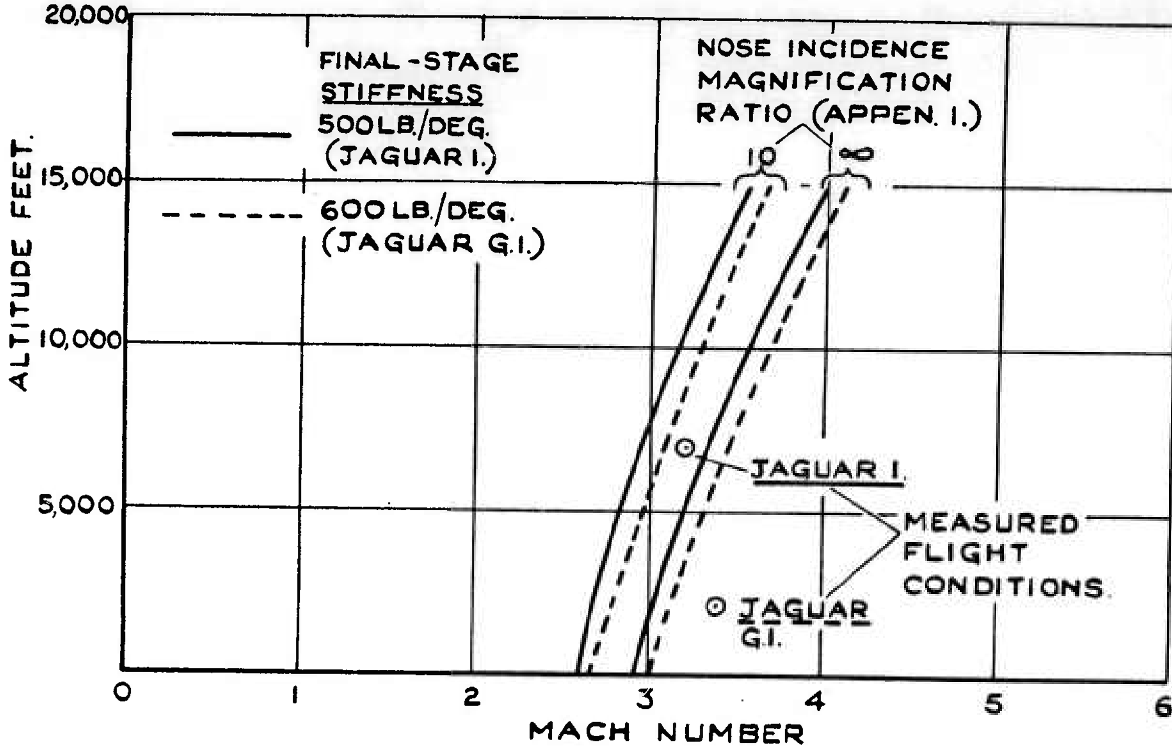


Fig.15 Estimated aeroelastic effects at the end of first-stage burning

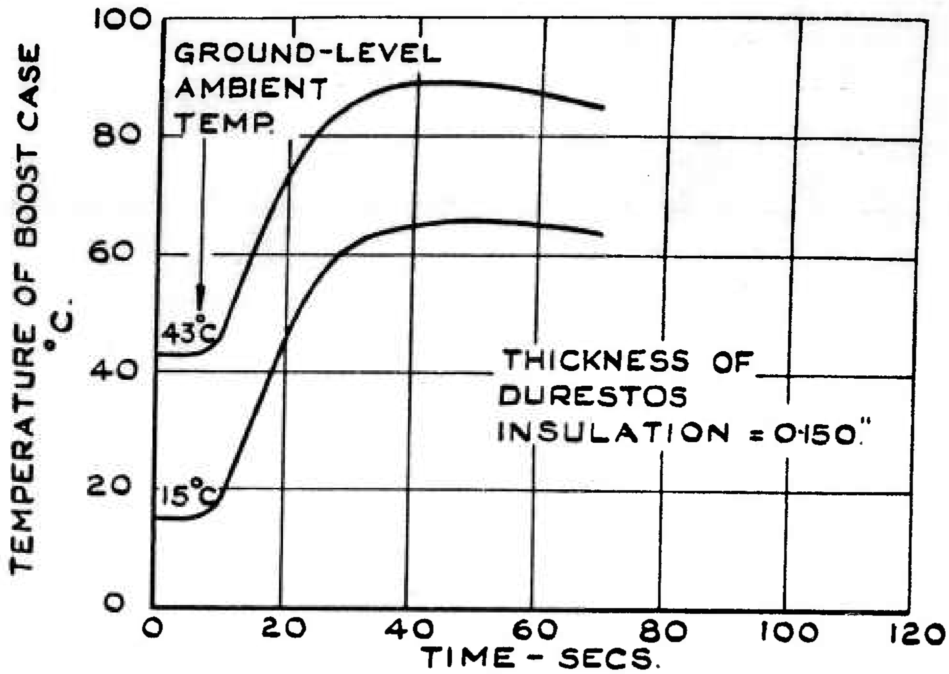


Fig. 16 Variation of temperature of second-stage boost case with time

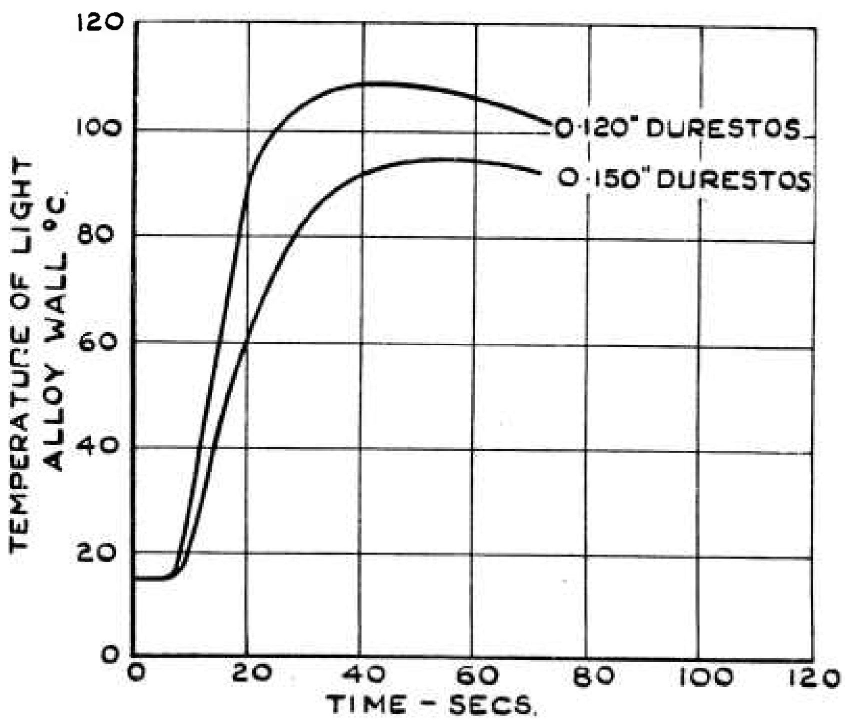


Fig.17 Variation of temperature on second-stage flare

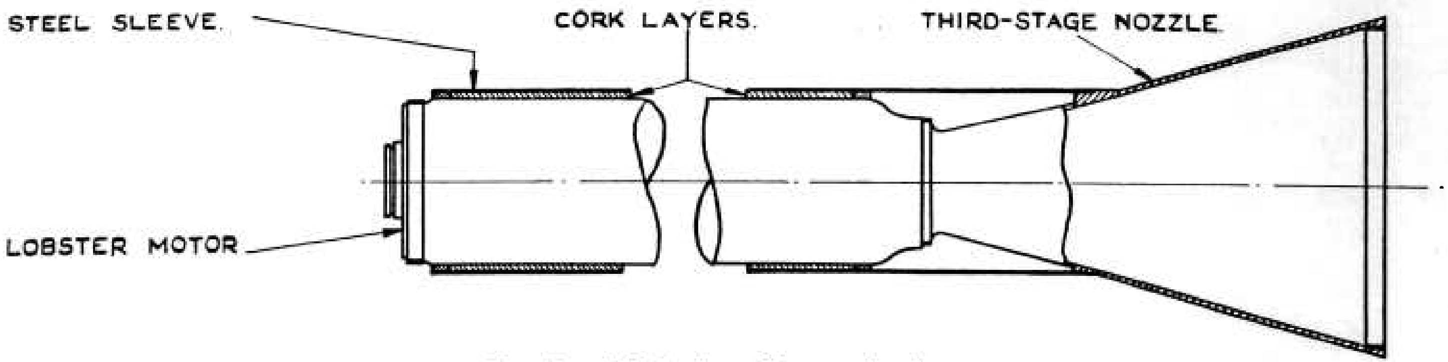


Fig.18 Third-stage flare and motor

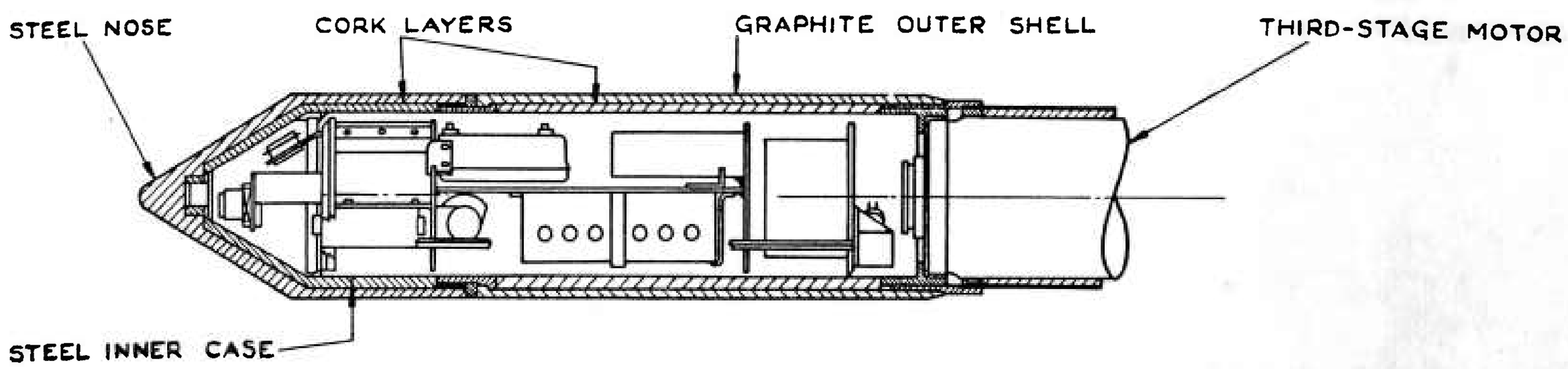


Fig.19 Instrument bay

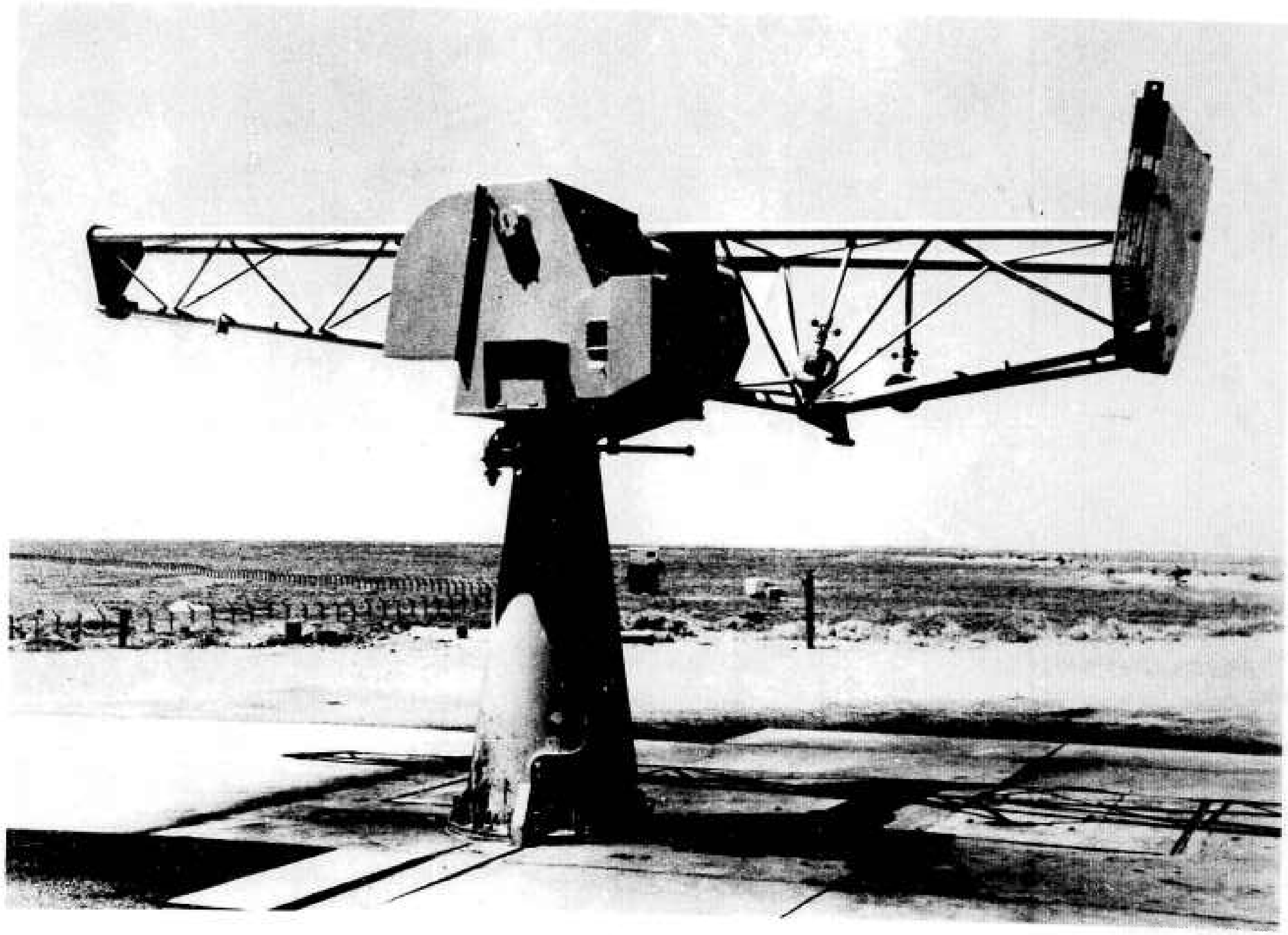


Fig.20 Zero-length launcher

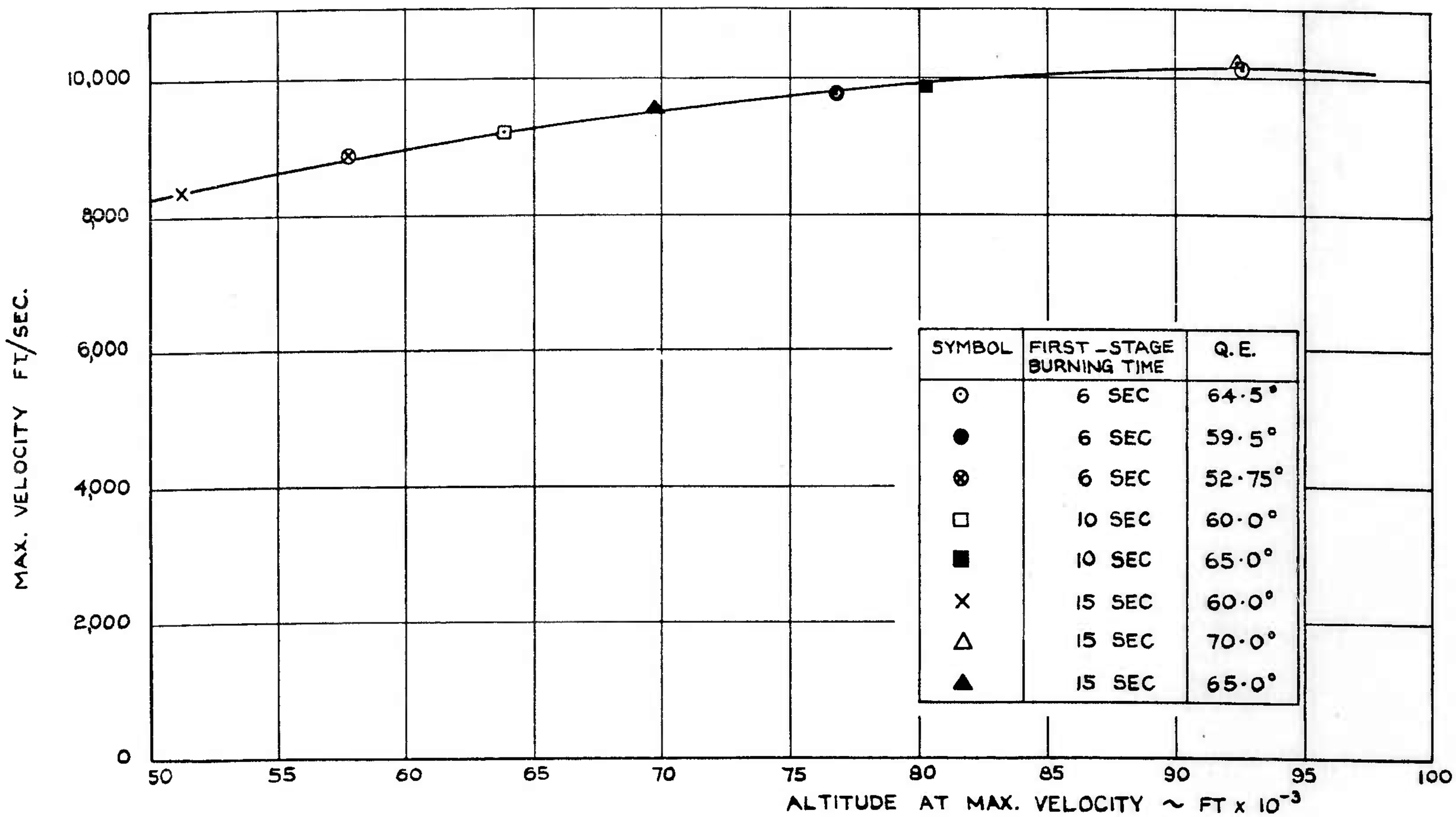


Fig.21 Altitude/velocity performance with various first-stage motors (estimated)

APPENDIX

DIVERGENCE OF A FLEXIBLE THREE-STAGE VEHICLE

Derivation of Expressions for the Trim and Nose Incidence Ratios, and the Divergence Parameters, by a Simple Static Analysis

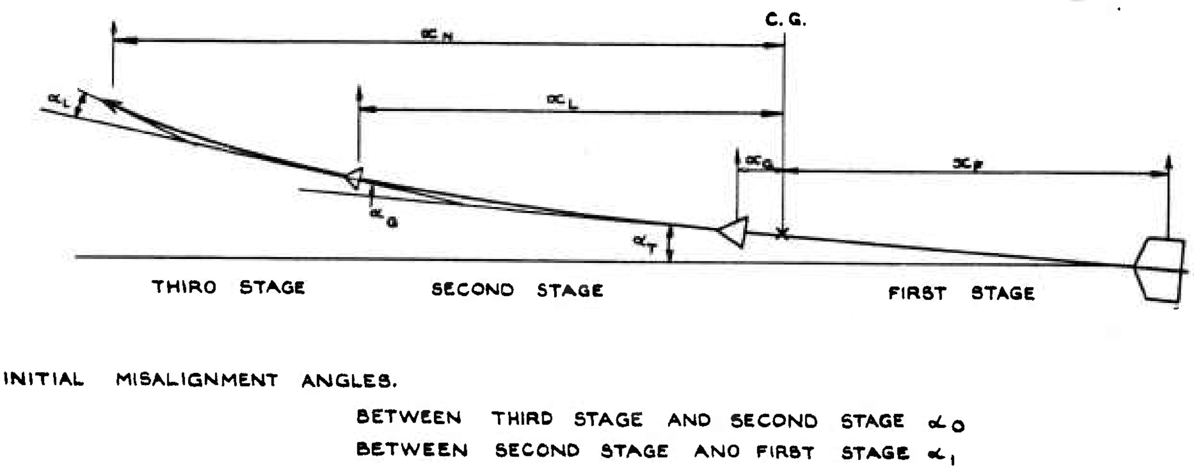
Assumptions

(i) The second and third stages only are flexible so the interstage couplings and the first stage are assumed to be rigid.

(ii) The vehicle trims out under steady conditions.

(iii) The third stage is initially misaligned to the second at an angle α_0 and the second stage is initially misaligned to the first stage at an angle α_1 .

(iv) All dynamic and inertia terms are ignored.



Notation (See Figure A1)

α_L = incidence of 3rd stage nose to 2nd stage nose

α_G = incidence of 2nd stage nose to the 1st stage nose

α_T = trim incidence of the 1st stage

K_L = stiffness of 3rd stage: load at nose for unit nose deflection (lb/degree)

K_G = stiffness of 2nd stage: load at its nose for unit deflection (lb/degree)

l_2, l_3 = effective cantilever lengths of 2nd and 3rd stages

a_N, a_L, a_G, a_F = lift-curve slope of 3rd stage nose, 3rd stage flare, 2nd stage flare and 1st stage fins respectively

S_N, S_L, S_G, S_F = reference areas corresponding to the above lift-curve slopes

X_N, X_L, X_G, X_F = distances from the centre of pressures of the lifting components to the centre of gravity of the whole vehicle, all measured positively

q = dynamic pressure (lb/ft^2)

From these definitions, the incidence of the 3rd stage nose to the airstream is $\alpha_L + \alpha_G + \alpha_T$ and the incidence of the 3rd stage flare is $\alpha_G + \alpha_T$, and of the 2nd stage flare and 1st stage fins is α_T.

The angular elastic deflection of the 3rd stage is $\alpha_L - \alpha_0$ and of the 2nd stage $\alpha_G - \alpha_1$.

Thus we have three unknowns, α_L, α_G and α_T, which can be expressed as functions of the initial misalignments, α_0 and α_1, using the following equations.

Equilibrium of forces acting on 3rd stage:

$$a_N S_N q(\alpha_L + \alpha_G + \alpha_T) = K_L(\alpha_L - \alpha_0) \tag{A.1}$$

Equilibrium of forces acting on 2nd stage:

$$\left(1 + \frac{2l_3}{l_2}\right) a_N S_N q(\alpha_L + \alpha_G + \alpha_T) + a_L S_L q(\alpha_G + \alpha_T) = K_G(\alpha_G - \alpha_1) \tag{A.2}$$

Equilibrium of moments on the whole vehicle under trimmed conditions:

$$X_N a_N S_N q(\alpha_L + \alpha_G + \alpha_T) + a_L S_L X_L q(\alpha_G + \alpha_T) + X_G a_G S_G q\alpha_T = X_F a_F S_F q\alpha_T \tag{A.3}$$

Rearranging (A.1) we get

$$\alpha_L = (\alpha_G + \alpha_T)\frac{P}{1 - P} + \frac{\alpha_0}{1 - P} \tag{A.4}$$

where $$P = \frac{a_N S_N q}{K_L}$$

Substituting in (A.2) for α_L from (A.4) and rearranging gives

$$\alpha_G = \frac{\dfrac{S}{1 - P}(\alpha_0 + \alpha_T) + t\alpha_T + \alpha_1}{1 - t - \dfrac{S}{1 - P}} \tag{A.5}$$

where $$S = \left(1 + \frac{2l_3}{l_2}\right)\frac{a_N S_N q}{K_G} \quad \text{and} \quad t = \frac{a_L S_L q}{K_G}$$

Equation (A.3) can be simplified in appearance by introducing the notation

$$N = a_N S_N q X_N , \quad L = a_L S_L q X_L , \quad F = a_F S_F q X_F , \quad G = a_G S_G q X_G$$

and written as

$$N(\alpha_L + \alpha_G + \alpha_T) + L(\alpha_G + \alpha_T) = (F - G)\,\alpha_T \tag{A.6}$$

Substitution for α_L and α_G from (A.5) and (A.4) into (A.6), gives, after some manipulation,

$$\frac{\alpha_T}{\alpha_0} = \frac{N(1 - t) + LS + (\alpha_1/\alpha_0)\,\{L(1 - P) + N\}}{(F - G)\,\{(1 - t)(1 - P) - S\} - L(1 - P) - N} \tag{A.7}$$

This is the 'trim incidence magnification ratio'. For computational purposes a value for the initial misalignment ratio α_1/α_0 has been assumed, namely 0.5.

The 'nose incidence magnification ratio' $(\alpha_L + \alpha_G + \alpha_T)/(\alpha_0)$ can then be obtained and is

$$\frac{\alpha_L + \alpha_G + \alpha_T}{\alpha_0} = \frac{(\alpha_1/\alpha_0) + 1 - t + (\alpha_T/\alpha_0)}{(1 - t)(1 - P) - S}$$

Divergence occurs when the denominator of (A.7) becomes zero, i.e. when

$$(F - G)\{(1 - t)\,(1 - P) - S\} - L(1 - P) - N = 0.$$

DISCUSSION

G.Y. Fokkinga (Netherlands): From Figure 12 it is not clear if, after burn-out of the first stage, separation between first and second stage will occur. If this is not the case, is the spigot then in some way refrained from withdrawing from the venturi of the second stage?

Can it be deduced from Figure 3 that, at the moment of igniting the second stage, the speed is already quite low? One would deduce this from the marked increase in the radius at the end of the coasting period.

Author's reply: Stages one and two are pinned together in such a way as to inhibit separation until second-stage ignition. When the second stage is fired the pins simply fail in tension, thus allowing the first stage to fall clear. This procedure is adopted to aid optical tracking during the coasting period after first-stage burn-out.

The velocity at second-stage ignition is about 1000 ft/sec.

J.G. Thibodaux (U.S.): Do you feel that it is really necessary to insulate the second-stage rocket motor case?

Author's reply: The necessity for insulation of the second stage arises primarily from the short burning time of the first stage. This results in the achievement of high velocity at low altitude, and hence higher heating rates than are associated with most NASA multi-stage vehicles having relatively long first-stage burning times. Another contributing factor is the need to keep the temperature of the bonding material between motor charge and motor case to a value less than 90°C.

A D D E N D U M

AGARD SPECIALISTS' MEETING

ON

"THE USE OF ROCKET VEHICLES IN FLIGHT RESEARCH"

List of Papers Presented

Following is a list of the titles and authors, together with the AGARD Report number, of twenty three papers presented at the above Meeting held at Scheveningen, Holland, in July 1961.

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Organisation du Traite de l'Atlantique Nord
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North Atlantic Treaty Organization, Advisory Group for Aeronautical Research and Development
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J.A. Hamilton
1961
27 pages incl. 4 refs., 21 figs; plus discussion and bibliography of papers presented at the 'Use of Rocket Vehicles in Flight Research' Meeting

The Weapons Research Establishment in Australia and the Royal Aircraft Establishment in England have jointly sponsored and developed a three-stage rocket test vehicle having as its primary function the extension of free-flight aerodynamic researches up to speeds of 10,000 ft/sec. So far, two vehicles have been flown successfully.

P.T.O.

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